Demonstration of the NSTAR Ion Propulsion System on the Deep Space One Mission *

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Deep Space 1 is the first interplanetary spacecraft to use an ion propulsion system for the primary delta-v maneuvers. The purpose of the mission is to validate a number of technologies, including ion propulsion and a high degree of spacecraft autonomy, on a flyby of an asteroid and a comet. The ion propulsion system has operated for a total of over 14,200 hours at engine power levels ranging from 0.48 to 1.94 kW and has completed the encounter with the asteroid 1992KD and the comet Borrelly. The system has worked extremely well after an initial grid short was cleared after launch. Operation on the DS 1 spacecraft has demonstrated all ion propulsion system and autonomous navigation functions. All propulsion system operating parameters are very close to the expected values with the exception of the thrust at higher power levels, which is about 2 percent lower than that calculated from the electrical parameters. This paper provides an overview of the system performance from the first 14,200 hours of ion propulsion system operation in interplanetary space.

Introduction

NASA's New Millennium Program (NMP) is designed to flight validate high risk, high payoff technologies that will be required to execute future Earth orbital and Solar System exploration missions. A xenon ion primary propulsion system (IPS) is one of the key technologies being demonstrated on Deep Space 1 (DS1), the first of the New Millenium missions [?]. This spacecraft was launched in October, 1998, flew by the asteroid Braille (1992KD) in July, 1999 and the comet Borrelly in September, 2001. The validation objectives of DS1 include demonstrating the functionality and performance of the ion propulsion system in an environment similar to what will be encountered by future users, the compatibility of the IPS with the spacecraft and science instruments, and autonomous navigation and control of the IPS with minimum ground mission operations support. The in-space performance of the propellant feed system is discussed in reference [?] and preliminary results on the interactions of the IPS with the spacecraft and instrumentation are presented in [?]. This paper summarizes the results of validation activities associated with the engine performance and mission operations for the first 14,200 hours of engine operation.

Overview of the NSTAR Ion Propulsion System

The flight ion propulsion system was delivered to DS1 by the NASA Solar Electric Propulsion (SEP) Technology Applications Readiness (NSTAR) program, a joint Jet Propulsion Laboratory/Glenn Research Center effort to develop NASA's 30 cm ion

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Table 1: Flight throttle table of parameters controlled by the DCIU.

NSTAR Throttle Level	Mission Throttle Level	Beam Supply Voltage (V)	Beam Supply Current (A)	Accelerator Grid Voltage (V)	Neutralizer Keeper Current (A)	Main Plenum Pressure (psia)	Cathode Plenum Pressure (psia)
15	111	1100	1.76	-180	1.5	87.55	50.21
14	104	1100	1.67	-180	1.5	84.72	47.50
13	97	1100	1.58	-180	1.5	81.85	45.18
12	90	1100	1.49	-180	1.5	79.29	43.80
11	83	1100	1.40	-180	1.5	76.06	42.38
10	76	1100	1.30	-180	1.5	72.90	41.03
9	69	1100	1.20	-180	1.5	69.80	40.26
8	62	1100	1.10	-180	1.5	65.75	40.26
7	55	1100	1.00	-150	2.0	61.70	40.26
6	48	1100	0.91	-150	2.0	57.31	40.26
5	41	1100	0.81	-150	2.0	52.86	40.26
4	34	1100	0.71	-150	2.0	48.08	40.26
3	27	1100	0.61	-150	2.0	43.18	40.26
2	20	1100	0.52	-150	2.0	39.22	40.26
1	13	850	0.53	-150	2.0	39.41	40.26
0	6	650	0.51	-150	2.0	40.01	40.26

Table 2: Flight throttle table of parameters used in mission analysis.

NSTAR Throttle Level	Mission Throttle Level	PPU Input Power (kW)	Engine Input Power (kW)	Calculated Thrust (mN)	Main Flow Rate (sccm)	Cathode Flow Rate (sccm)	Neutralizer Flow Rate (sccm)	Specific Impulse (s)	Total Efficiency
15	111	2.52	2.29	92.4	23.43	3.70	3.59	3120	0.618
14	104	2.38	2.17	87.6	22.19	3.35	3.25	3157	0.624
13	97	2.25	2.06	82.9	20.95	3.06	2.97	3185	0.630
12	90	2.11	1.94	78.2	19.86	2.89	2.80	3174	0.628
11	83	1.98	1.82	73.4	18.51	2.72	2.64	3189	0.631
10	76	1.84	1.70	68.2	17.22	2.56	2.48	3177	0.626
9	69	1.70	1.57	63.0	15.98	2.47	2.39	3136	0.618
8	62	1.56	1.44	57.8	14.41	2.47	2.39	3109	0.611
7	55	1.44	1.33	52.5	12.90	2.47	2.39	3067	0.596
6	48	1.32	1.21	47.7	11.33	2.47	2.39	3058	0.590
5	41	1.19	1.09	42.5	9.82	2.47	2.39	3002	0.574
4	34	1.06	0.97	37.2	8.30	2.47	2.39	2935	0.554
3	27	0.93	0.85	32.0	6.85	2.47	2.39	2836	0.527
2	20	0.81	0.74	27.4	5.77	2.47	2.39	2671	0.487
1	13	0.67	0.60	24.5	5.82	2.47	2.39	2376	0.472
0	6	0.53	0.47	20.6	5.98	2.47	2.39	1972	0.420

thruster technology for flight applications with industry participation from Hughes Electron Dynamics, Moog, Inc. and Spectrum Astro, Inc. The ion thruster uses propellant delivered by a Xenon Feed System (XFS) and is powered by a Power Processing Unit (PPU), which converts power from the solar array to the currents and voltages required by the engine. The XFS and PPU are controlled by a Digital Control Interface Unit (DCIU), which accepts and executes high level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system. Planetary missions often require a wide throttling range to accommodate variations in solar array output power with distance from the Sun, so the NSTAR IPS was designed to operate over an engine power range of

0.5 kWe to 2.3 kWe. The development of the flight system is discussed in detail in references [?, ?].

The NSTAR Throttle Table

The DCIU is capable of operating the thruster at any one of 16 discrete throttle levels from a throttling table stored in memory. This table contains the setpoints for the PPU power supplies and the XFS pressures and can be modified by ground command. The NSTAR 16 level throttle table showing the entire range of operation is listed in Table (1). The NSTAR 16 point throttle table contains the IPS setpoints required to operate the system over the required throttling range. A corresponding mission throttle table containing the flow rates, thrust and PPU input and output power levels is required to perform the mission trajectory design. The NSTAR mission table is listed in Table (2). The development of these throttle tables is described in this section.

Power throttling is accomplished by varying the beam voltage and current. The engine throttling envelope with lines of constant beam power is shown in Fig. (1). The boundaries of this envelope represent the maximum beam voltage and current capabilities, the minimum beam current (which is determined primarily by the minimum discharge current) and the beam voltage perveance limit. The NSTAR throttle table was designed to maximize the specific impulse, so the power is varied with beam current throttling over most of the range. The lowest power levels are achieved by operating at the minimum beam current and throttling the beam voltage.

The discharge chamber flow rate was selected to give the propellant utilization shown in Fig. (2). The propellant efficiency of 0.9 was chosen at high power levels as a compromise between maximizing total engine efficiency and minimizing double ion production, which can drive internal erosion rates. A propellant efficiency of 0.9–0.91 is maintained over most of the range. At the lowest powers the double-to-single ion current ratio is low, so lower propellant efficiencies were chosen to optimize performance and achieve the desired total power.

The neutralizer and cathode flow rates are approximately equal at each operating point and vary over the throttling range as shown in Fig. (3). The minimum flow rate was designed to prevent the neutralizer from operating in plume mode, which can cause excessive erosion of the orifice. End-of-life neutralizer characterization data from the 8200 hour Life Demonstration Test (LDT) of an engineering model thruster (EMT2) are shown on this plot as well [?]. The maximum flow rate was chosen to match the discharge cathode flow rate used in a 1000 hour test of an engineering model thruster which demonstrated little cathode erosion compared to a previous 2000 hour test at a lower flow rate [?]. Subsequent tests suggest that other design changes were responsible for the erosion rate reduction, but the higher flow was maintained to be conservative.

The thrust in the mission throttle table is calculated from the engine electrical setpoints,

$$T = \alpha F_t J_b (V_s - V_g)^{1/2} \left(\frac{2M}{e}\right)^{1/2} \qquad (1)$$

where J_b is the beam current, V_s is the beam power supply voltage, V_g is the coupling voltage between neutralizer common and the facility ground or ambient space plasma, M is the mass of a xenon ion and e is the charge of an electron. The factors α and F_t correct for the doubly-charged ion content of the beam and thrust loss due to beam divergence [?]. A constant value of 0.98 for F_t based on earlier 30-cm thruster ground tests and a value of α based on a curve fit to centerline double ion current measurements as a function of propellant utilization efficiency in a 30 cm, ring-cusp inert gas thruster [?] were used. Earlier direct measurements of thrust from the LDT agreed well with the calculated value [?, ?]. More recent measurements with the flight thrusters were somewhat lower than the calculated values for intermediate throttle levels. The difference between the measured thrust and the table values is shown in Fig. (4).

The power required for a given thrust level increases over the engine lifetime due to wear [?], so two tables representing beginning-of-life (BOL) and end-of-life (EOL) were developed. These have the same engine setpoints shown in Table (1) but different engine power levels. The BOL table was developed primarily through testing with engineering

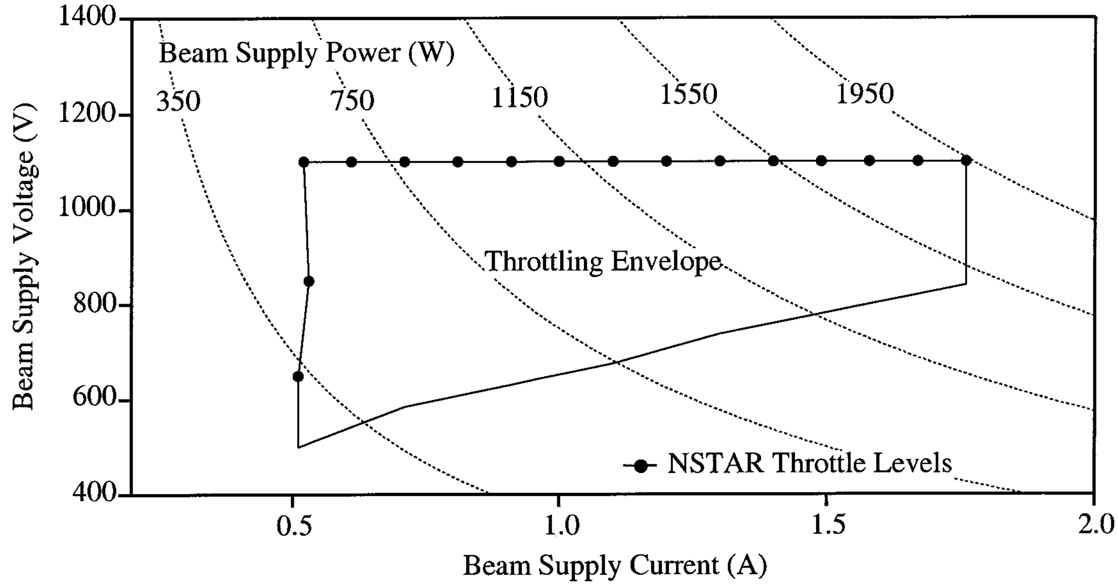

Figure 1: NSTAR power throttling strategy.

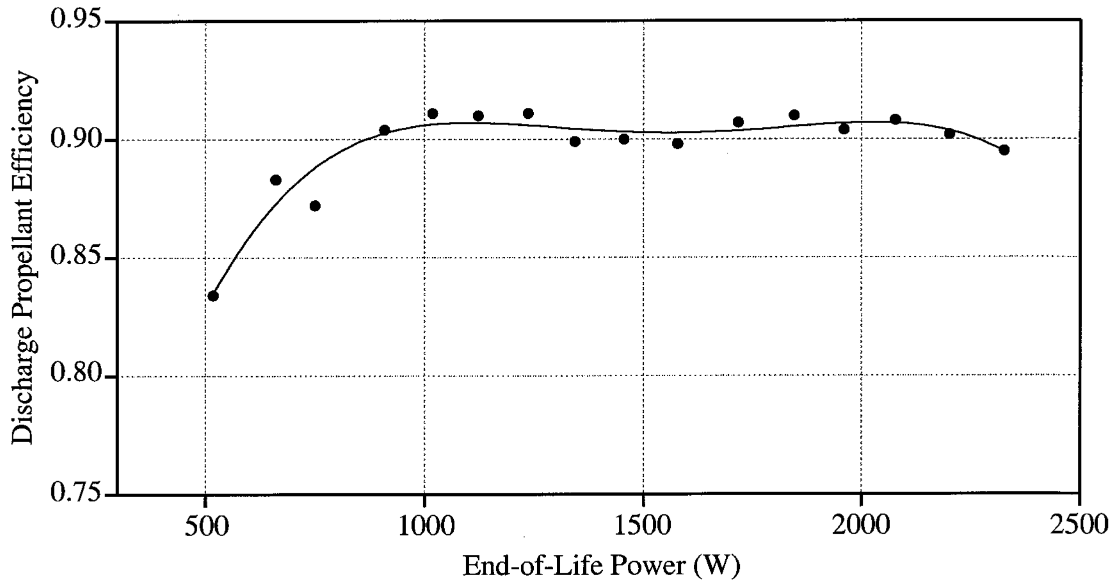

Figure 2: NSTAR ion thruster discharge propellant utilization efficiency.

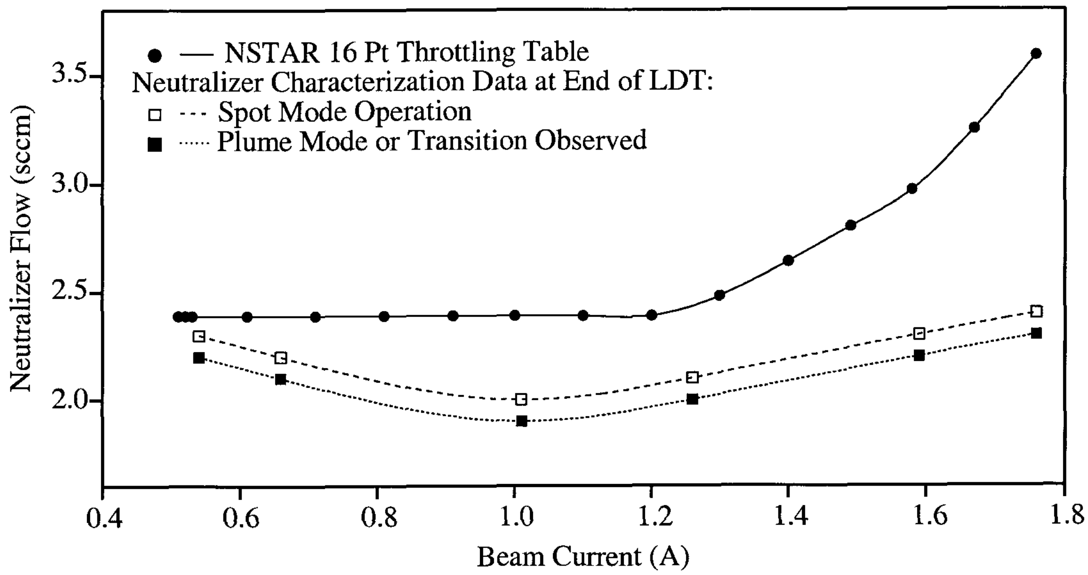

Figure 3: NSTAR cathode flow rates.

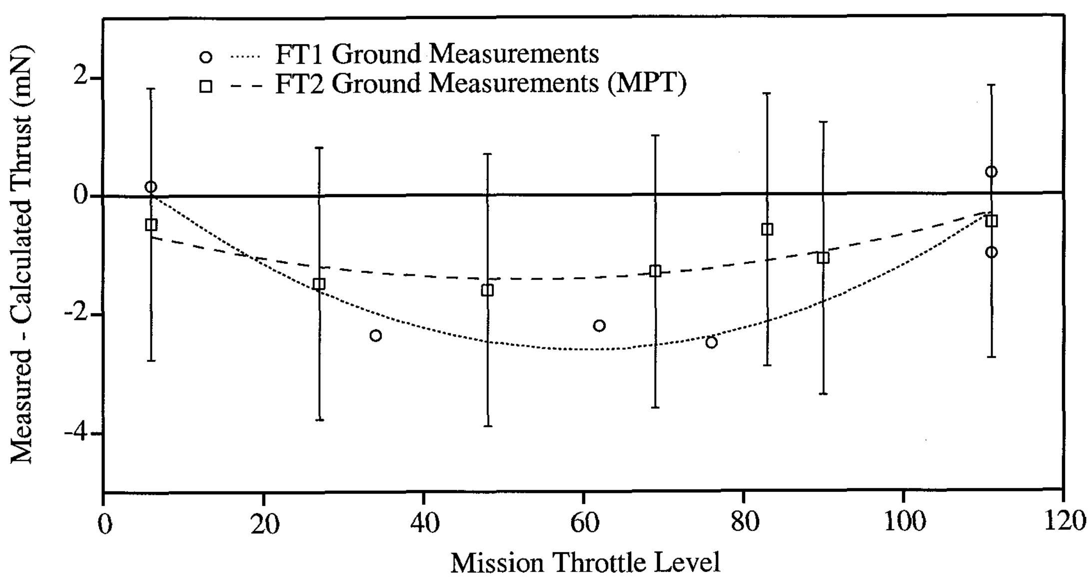

Figure 4: The difference between measured and calculated thrust over the NSTAR throttling range.

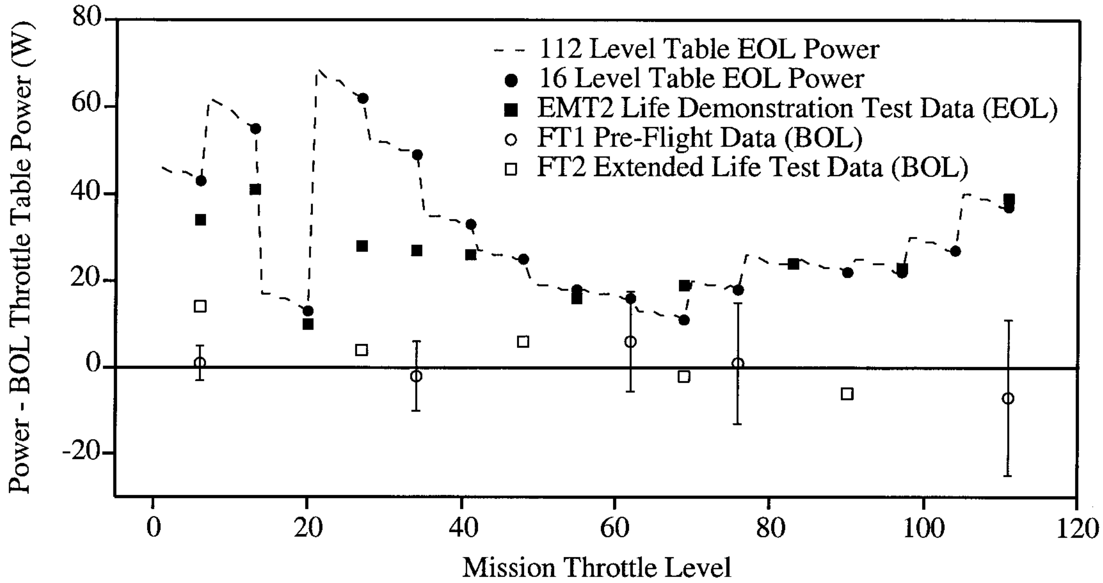

Figure 5: Difference between a given power level and the beginning-of-life power.

model thrusters and updated with data from pre-flight measurements with FT1. The EOL table was based largely on measurements from the 8200 hour test of EMT2. The power at the lowest throttle levels was extrapolated from performance curves obtained after about 6500 hours of operation. The extrapolations were based on sensitivity data, which were used to correct for slight differences in some of the controlled parameters. The difference between BOL and EOL engine power is plotted in Fig. (5). Additional measurements taken at some of these throttle levels after about 6900 hours of operation in the LDT are also shown. They suggest that the EOL power at some of the lower throttle levels is overestimated in the throttling table. BOL data obtained with the two flight thrusters demonstrates that their initial performance agrees well with the table values.

The PPU input power corresponding to a given engine power is determined by the PPU efficiency. The efficiency of the flight PPU was characterized as a function of input bus voltage and temperature in several ground tests, as shown in Fig. (6). The lowest measured values over this range of parameters were used to define the lowermost line in the figure. This conservative estimate of PPU efficiency was used to generate the PPU input powers in the throttle table.

To make finer steps in power throttling and more closely track the solar array peak power, a 112 point throttle table was also developed. Throttling between the 16 NSTAR throttle points is accomplished by varying the beam voltage to give steps approximately 20 W apart. A 16 point subset of this table is loaded into the DCIU to provide fine control over a restricted power range for a given mission phase.

In-Flight System Performance

One of the primary objectives of the flight validation activity is to verify that the system performs in space as it does on the ground. The parameters of interest to future mission planners are those in the mission throttle table: thrust and mass flow rate as a function of PPU input power. In this section the system power, thrust and mass flow rate behavior will be evaluated in terms of the throttle table.

XXX-figure out where to put this...The flight electrical telemetry is calibrated to within ± 2 percent of reading for the high voltage supply parameters and ± 2 percent of full scale for the other parameters. In this paper the values have been corrected using the ground calibration data and are more accurate–typically the standard error is under 0.2–0.8 percent of full scale. The voltage measurements have also

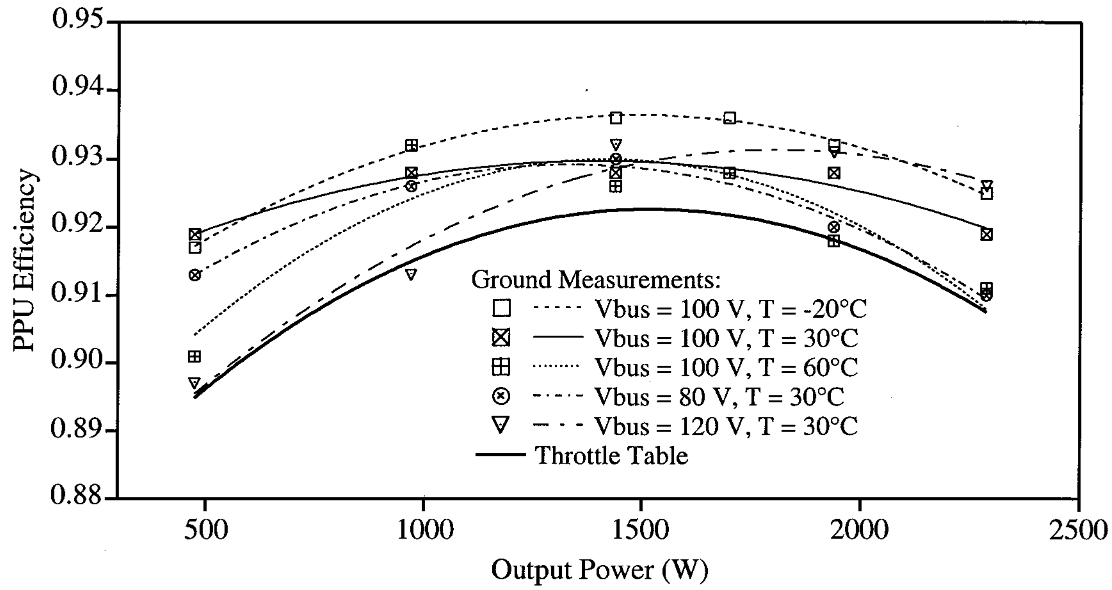

Figure 6: PPU efficiency measurements.

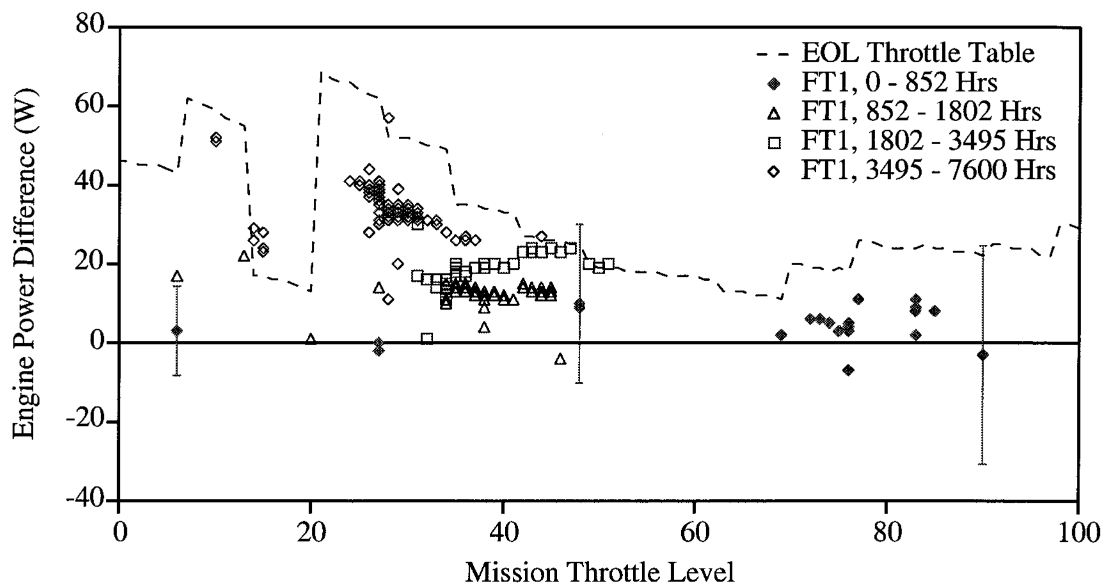

Figure 7: Difference between a given engine power level and the throttle table BOL values.

been corrected for flight cable line drops and represent the values that would be measured at the engine.

PPU Power Input Requirements

The PPU input power is determined by the PPU output power (engine power requirement) and the PPU efficiency. The difference between the in-flight engine input power and the BOL throttle table power is shown in Fig. (7). These power values are based on the individual power supply current and voltage telemetry readings. The total engine power consumed during the IAT1 throttle test and initial operations differed from the table values by only about 2 W on average, although the uncertainties are much larger than this, as shown by representative error bars on the figure. The engine power requirement increased by 12–15 W with time, however, as indicated by the data from the second major thrust period from 852 to 1802 hours of operation. During the third major thrust arc from 1802 to 3495 hours the engine power at throttle levels between 40 and 50 has increased to the EOL power values used in the throttle table, which is represented by the dashed line in Fig. (7). The last 4100 hours of operation have been primarily at lower power levels. At mission throttle levels between 25 and 40 the engine power has increased by 30 to 40 W, but is still not as high as the EOL throttle table values. More recent operation at levels 15 and 10 for thrust vector control suggests that at the lowest power levels the engine is now operating near the EOL throttle table values. This increased power demand is due primarily to increased discharge power losses, as discussed in the next section. This is a normal consequence of engine aging [?, ?], but does not imply that the engine is nearing end of life. In the 8200 hour Life Demonstration Test the engine power increased during the first 3000 hours and was relatively constant thereafter [?].

In-flight measurements of the PPU efficiency suggest that it is higher than the conservative value assumed in the throttle tables, as shown in Fig. (8). These values are based on the total engine power and PPU high voltage bus current and voltage telemetry with an additional 15 W assumed for the low voltage bus input power. There is no telemetry for the low voltage bus, but ground testing showed a 15 W loss for all conditions. The efficiency is sensitive to the line voltage and the temperature, as the ground data show. The in-flight measurements should be compared with the solid line in the center of the preflight data and the highest dashed line. The range of uncertainty in these measurements encompasses the ground test data, but the in-space measurements appear to be comparable to or higher than the ground measurements.

Because the PPU efficiency is higher than planned in the mission throttle table, it more than offsets the increased output power requirements observed so far in the actual flight. Figure (9) displays the difference between the observed PPU input power and the BOL input power from the throttle table. The input power required early in the mission was approximatly 20 W lower than expected, because of the higher PPU efficiency. The data from the subsequent NBURNs show that the input power is just now approaching the BOL throttle table value.

IPS Thrust

The acceleration of the spacecraft is measured very accurately from changes in the Doppler shift of the telecommunications signals. With models of the spacecraft mass as a function of time, the Doppler residual data can be used to measure the thrust of the IPS with an uncertainty of less than 0.5 mN [?]. Preliminary thrust measurements have been obtained so far in the first 852 hours of thrusting from IAT1 and during IAT2 at 1800 hours. The flight beam voltage and current values, which determine to a large extent what the thrust is, are slightly different from the setpoints in the table. The flight thrust measurements are therefore compared to the thrust calculated from the actual electrical parameters rather than the table values. The difference in the measured and calculated thrust is shown in Fig. (10), with the curve fits to similar data obtained with a thrust balance in ground tests. The ground and flight data agree well with the calculated values at low powers, but are lower at intermediate powers. The flight data suggest that the difference in true thrust and calculated thrust grows linearly with power and is up to 1.6 mN lower than

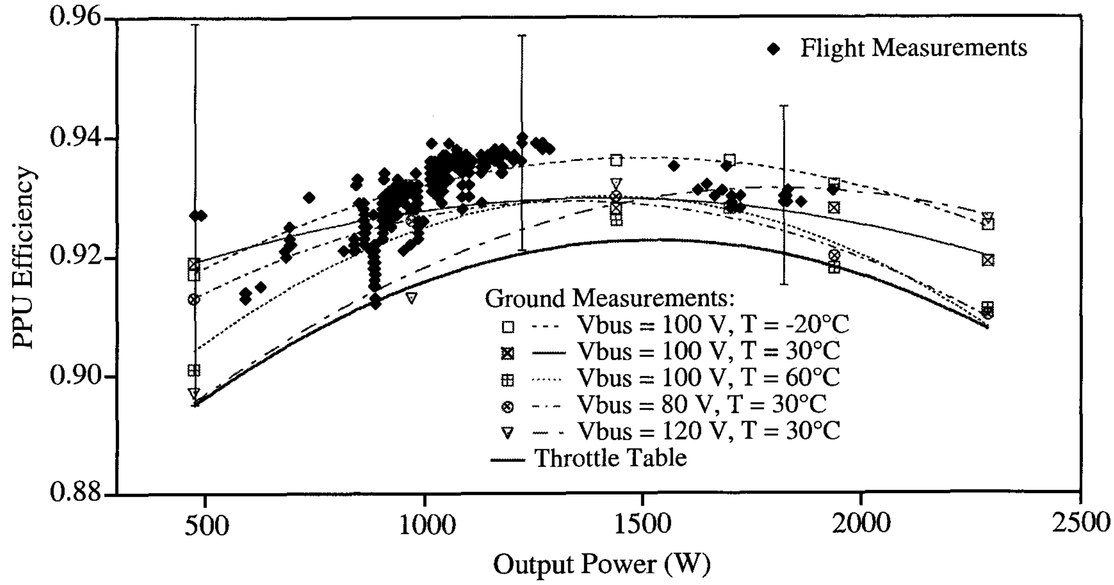

Figure 8: In-flight measurements of PPU efficiency compared to ground test data.

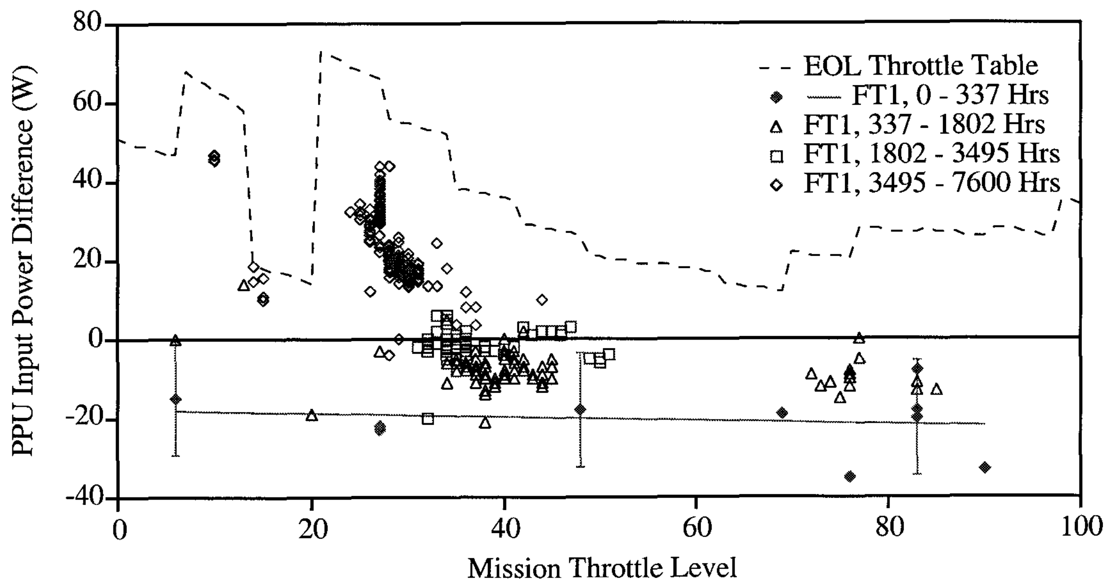

Figure 9: Difference between a given PPU input power level and the corresponding throttle table BOL value.

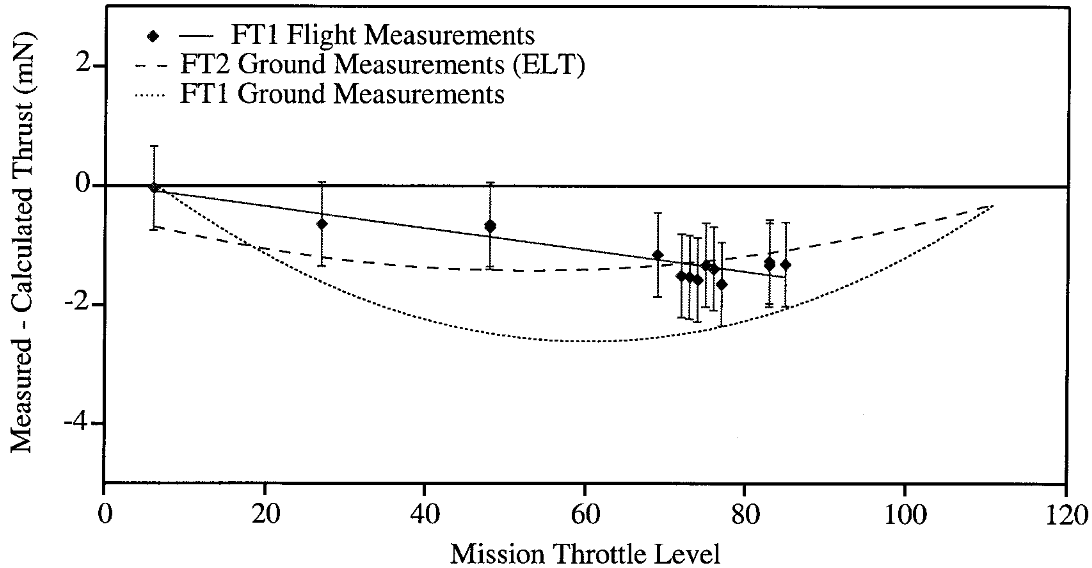

Figure 10: Difference between measured and calculated thrust in flight compared to ground measurements.

expected at mission level 83 (1.82 kWe engine power). The error bars are based on the uncertainty in the measured thrust and do not include errors in the calculated thrust.

This discrepancy may also be due to a systematic error in the flight telemetry, although the agreement with ground data argues against that conclusion. As Eq. (1) shows, the true thrust might be lower than calculated because of a higher double ion content, greater beam divergence than observed in the previous 30-cm thruster tests or differences in the coupling voltage in-space compared to ground tests. Additional measurements and analysis will be required to resolve this issue.

Although the actual thrust appears to be slightly lower than expected, at the beginning of the mission the overall system performance was still very close to the BOL throttle table level, in terms of thrust for a given PPU input power. Figure (11) shows that at the beginning of the mission the higher PPU efficiency largely compensated for the lower thrust. In this comparison, the thrust is within 0.5 mN of the table values. The gap between the two widens as the engine wears and the total engine power requirement for a given throttle level grows, however. The PPU input power required for the thrust levels measured during NBURN 0 has exceeded the EOL throttle table power for an equivalent thrust.

Propellant Flow Rates

The performance of the xenon feed system is discussed in detail in [?]. In general, the performance has been excellent, although the flow rates are slightly higher than the throttle table values. The mean value of the main flow is 0.05–0.14 sccm (about 0.4 to 1.0 percent) high, while that of the two cathode flows is 0.03 sccm (about 1.0 percent) high. This is in part intentional. As Fig. (12) shows, the XFS bang-bang regulators result in a sawtooth pressure profile. The control system is designed so that the minimum pressure in this sawtooth yields the throttle table flow rate values. In addition to this deliberate conservatism, there is a slight bias in both regulators because one of each of the three pressure transducers on the two plena had a slight offset after launch.

Overall System Performance

The propulsion system performance can be summarized in terms of specific impulse and efficiency. At the beginning of the mission the Isp was about 60 s

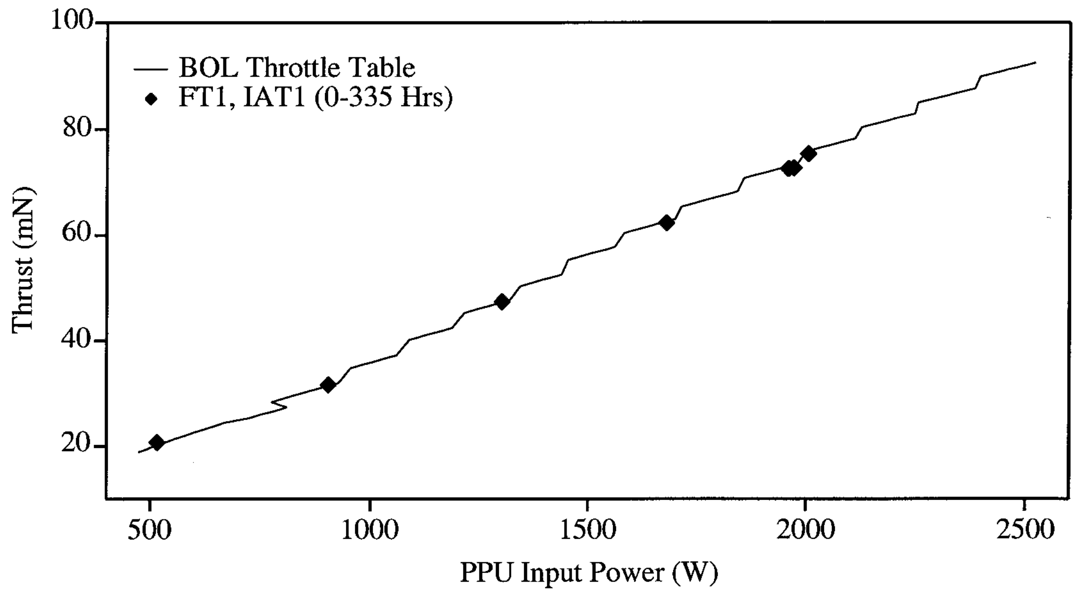

Figure 11: Measured thrust as a function of PPU input power compared to throttle table values.

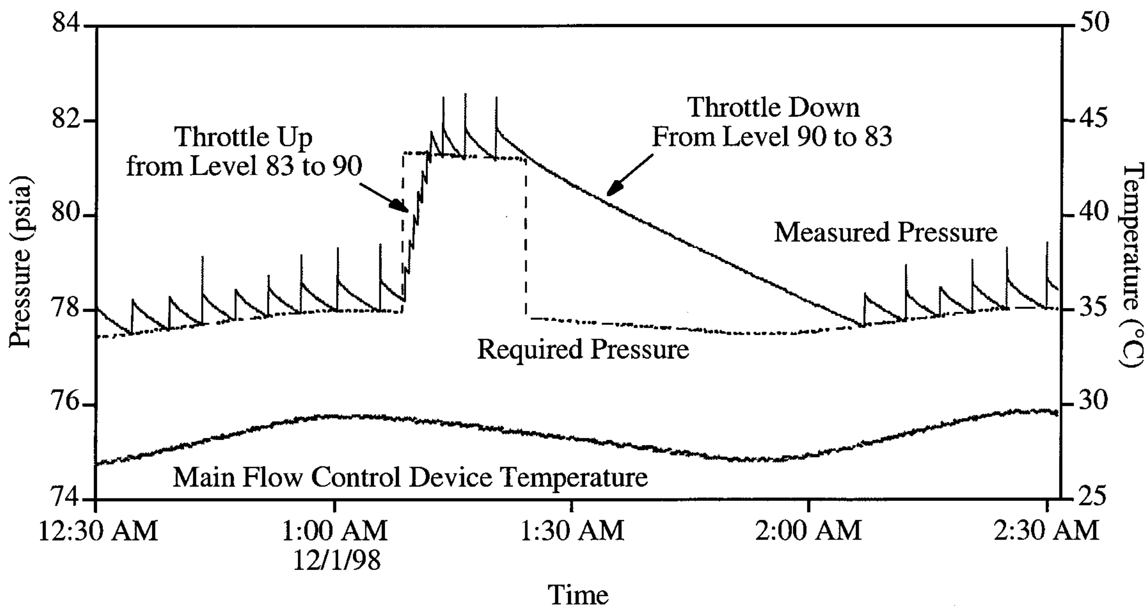

Figure 12: Example of flow rate throttling.

Table 3: Flight engine performance measured in space.

NSTAR Throttle Level	Mission Throttle Level	PPU Input Power (kW)	Engine Input Power (kW)	Measured Thrust (mN)	Main Flow Rate (sccm)	Cathode Flow Rate (sccm)	Neutralizer Flow Rate (sccm)	Specific Impulse (s)	Total Efficiency
Measurements from IAT1 at beginning of mission.									
0	6	0.501	0.478	20.797	6.05	2.50	2.43	1964	0.419
3	27	0.890	0.843	31.766	6.93	2.50	2.43	2776	0.513
6	48	1.286	1.222	47.298	11.42	2.50	2.42	2998	0.569
9	69	1.666	1.571	62.227	16.08	2.50	2.43	3068	0.596
11	83	1.944	1.823	72.561	18.63	2.75	2.67	3126	0.610
12	90	2.063	1.935	77.388	20.01	2.91	2.83	3114	0.611
Measurements from IAT2 after 1800 hours of operation.									
0	6	0.515	0.492	20.705	6.04	2.50	2.42	1958	0.404
1	13	0.670	0.626	24.234	5.88	2.49	2.41	2330	0.442
2	20	0.778	0.737	26.985	5.84	2.50	2.42	2598	0.467
3	27	0.910	0.860	31.460	6.91	2.51	2.43	2752	0.494
4	34	1.049	0.984	36.616	8.38	2.49	2.42	2854	0.521

lower than expected and the engine efficiency was 2 to 2.5 percentage points lower than the throttle table values. The measured performance was still excellent, with a measured efficiency of 0.42 to 0.60 at Isp's ranging from 1960 to 3125 s over an engine throttling range of 478 to 1935 W. The measured mission planning performance parameters are summarized in Table (3).

Engine Behavior In Flight

The engine behavior in space has been very similar to that observed in ground testing. The detailed operating characteristics of the engine are discussed in this section.

Engine Ignitions

A total of 103 successful engine ignitions have occurred in the first 7600 hours of the mission with only one failure to achieve beam extraction due to the initial grid short discussed above. The data from these ignitions are reviewed here. The nominal heater current value is 8.5 A; the actual cathode and neutralizer heater currents in-flight have been constant and within about 2 percent of the setpoint value. The uncertainty in these measurements is about ± 2 percent. The peak heater voltage is a function of the heater impedence, current and temperature. The flight telemetry shows that the heater voltage increases in any rapid sequence of ignitions because the conductor is hotter at the beginning of each consecutive start. The data show that the heater voltage is also higher when the initial thruster temperature is higher. The scatter in the peak voltages under similar temperature conditions is low and very similar to that observed in ground tests.

The time required for the cathodes to ignite after the 210 s heat phase and application of the high voltage ignitor pulses is also a function of the initial temperature, with 20–80 s delays in neutralizer ignition observed for the lowest temperatures. Delays of up to 86 s were also observed during ground thermal tests at the lowest temperatures [**?**], and are not considered to be a concern. The discharge cathode typically ignites 5–6 s after successful neutralizer ignition, which reflects delays in the start sequence. Its ignition delay may be shorter because it has a slightly higher heater current and because it automatically goes through a longer heat phase when the neutralizer ignition is delayed.

Throttling Characteristics

The throttling sequences were in all cases executed properly by the DCIU after receiving ground commands. An example of the throttling sequence is

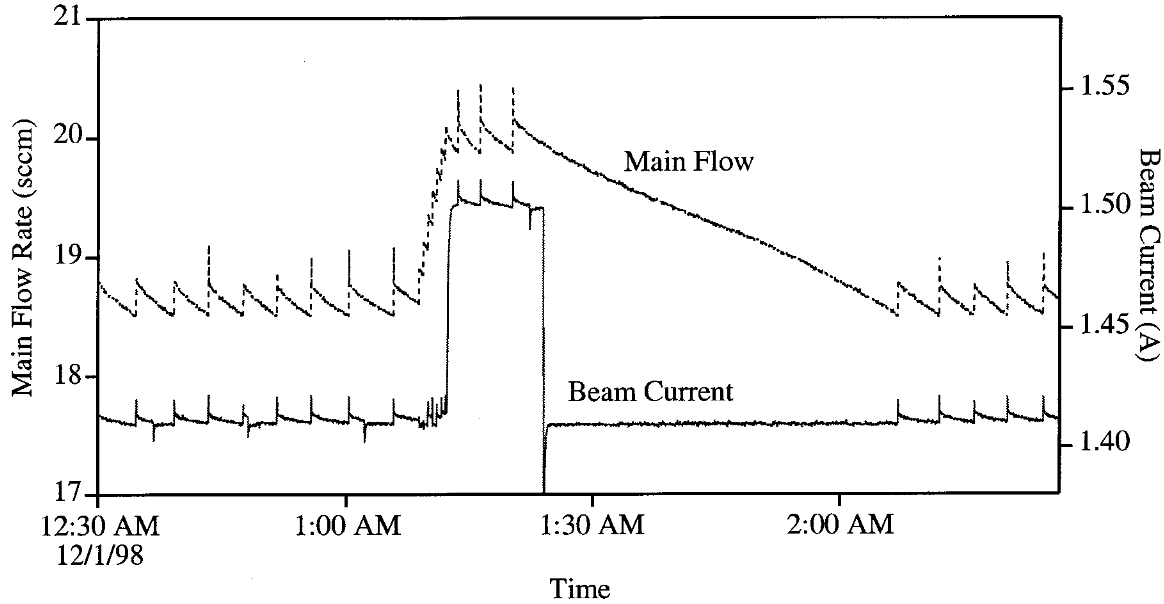

Figure 13: Example of throttle-up and throttle-down sequences.

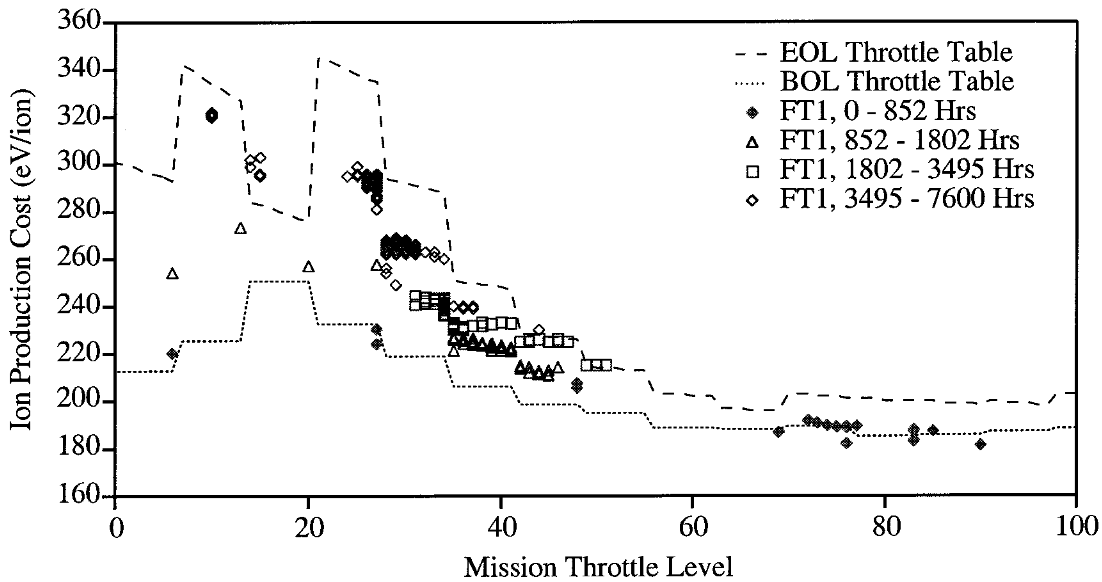

Figure 14: Discharge losses measured in flight compared to throttle table values.

shown in figures (12) and (13). The software onboard the spacecraft is also designed to autonomously throttle the engine and, if necessary, track the peak power available from the array. The Navigation Manager software recalculates the throttle level every 12 hours and commands the IPS to the proper throttle level via the IPS Manager. The Navigation Manager uses models of the solar array power, spacecraft power consumption and the trajectory to calculate the throttle level. If the solar array output power cannot supply the demands of the spacecraft and the IPS, power is drawn from an auxiliary battery. The spacecraft flight software battery algorithm will autonomously throttle the engine down if the battery discharge rate and charge drop below a prescribed threshold. This new throttle level will be maintained until the Navigation Manager resets the throttle level again. This function was successfully demonstrated in all of the NBURNs, which were accomplished with no ground control over the detailed engine operation required.

Steady-State Setpoint Accuracy

As mentioned above, the flight flow rates are slightly higher than the throttle table setpoints. In addition, the beam current is approximately 1 percent high over a range of 0.51 to 1.49 A. The beam current is controlled in flight to within ± 2 mA by varying the discharge current in closed-loop. Variation in the beam current is driven primarily by the flow rate sawtooth, as shown in Fig. (13). The neutralizer keeper current is within one percent of the setpoint. The accelerator grid voltage is 1.1 percent higher than the setpoint at the nominal full power operating point. The beam voltage is on average about 0.3 percent lower than the full power setpoint. The offsets in beam power supply settings result in slightly higher beam power levels than the throttling tables assume. This is largely offset by lower neutralizer power levels, as explained below. All of these parameters are well within the specified flight system tolerances.

Discharge Performance

As indicated in the previous section, the difference between the total engine power and the throttle table values is dominated by the discharge power difference. The discharge performance is summarized in terms of the ion energy cost in Fig. (14). The standard error of these measurements is 1.5 percent. This plot shows the beginning- and end-of-life discharge loss as a function of mission throttle level. The data from early in the DS1 mission are quite close to the throttle table values except in the middle of the range (throttle levels 40-60), where the flight data are higher. This appeared to be true of the ground measurements as well, suggesting that the BOL throttle table discharge power is low by about 10 W in this range. The data from the subsequent thrust periods indicate that the discharge losses are increasing with time as a consequence of engine wear [?, ?]. The lowest throttle levels are particularly sensitive to engine wear [?] and show the largest increases in flight, up to 95 W. The data from the thrust arc from 1802 to 3495 hours show that the discharge losses have increased to the EOL throttle table values in the mid-power range. During the last 4100 hours of operation the discharge loss has increased to the EOL values for the lowest power levels, although the losses at throttle levels between 25 and 40 are still below the EOL values.

The discharge voltage and current are compared with the throttle table values in figures (15) and (16). The voltages measured in flight are typically within 2 percent of the throttle table voltages. The ground test data are also plotted in this figure and tend to be slightly higher, although some of these measurements have not been corrected for voltage drops in the ground facility power cables. There is very little drift in the discharge voltage over the course of the flight, which is consistent with long duration ground test data [?, ?, ?]. The discharge current is also close to the BOL table values initially, with the exception of measurements at mission level 48. This is in the range where the table values appear to underestimate true BOL behavior. Unlike the voltage, the discharge current has increased with time driving the discharge power toward the EOL values.

Data on the sensitivity of discharge losses, voltage and current to small variations in flow rates and beam current from the on-going Extended Life Test

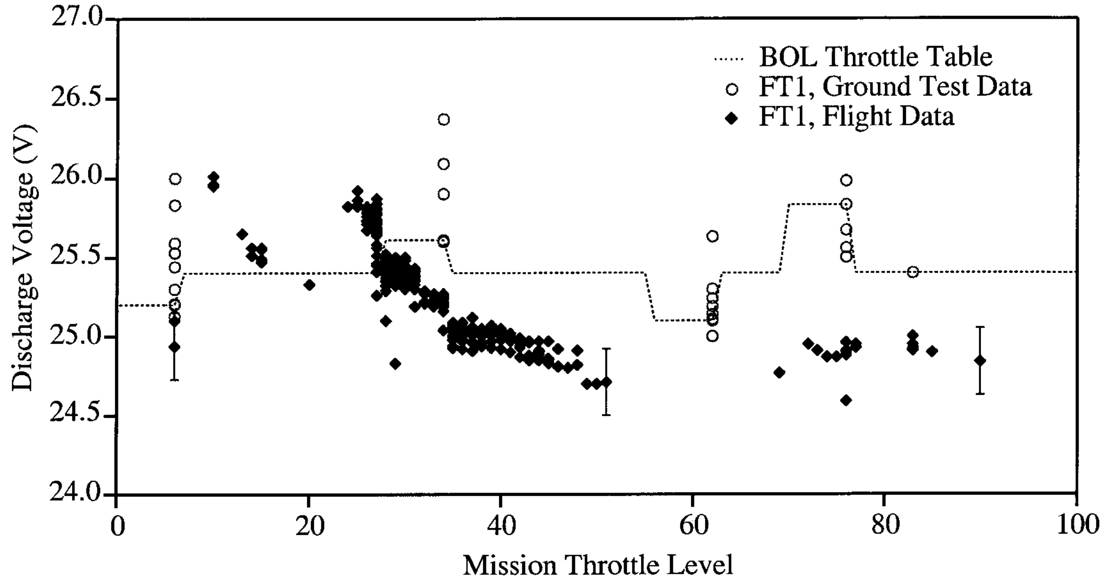

Figure 15: Discharge voltage measured in flight compared to throttle table and ground test measurements.

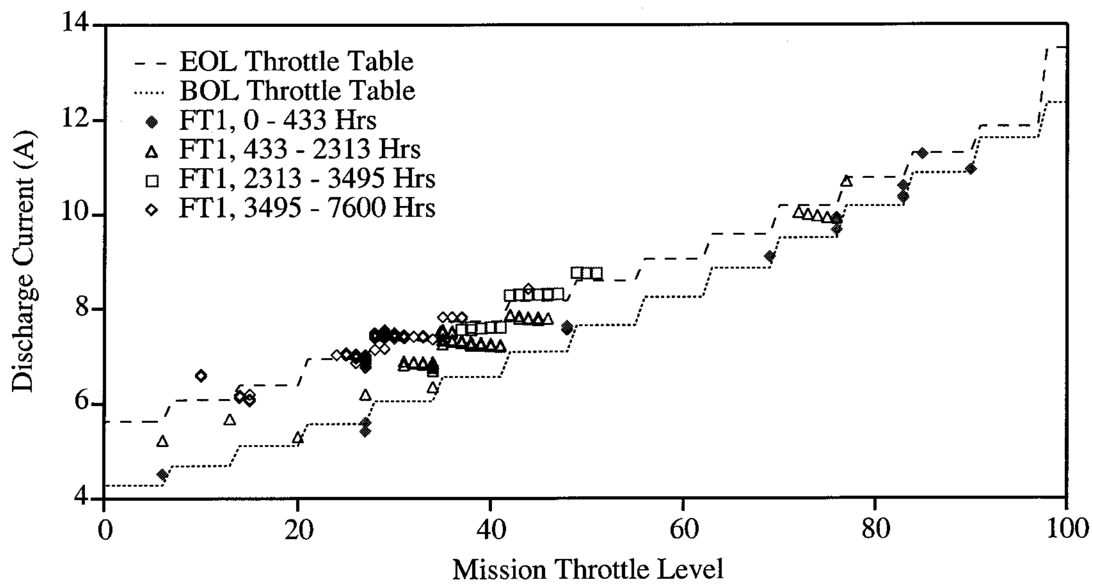

Figure 16: Discharge current measured in flight compared to throttle table values.

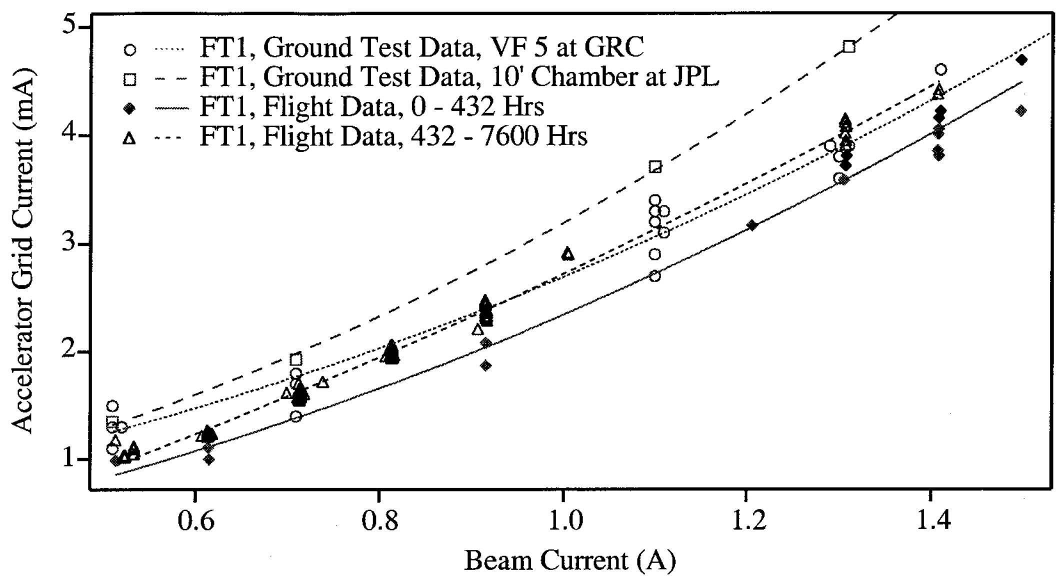

Figure 17: Accelerator grid impingement current measured in space compared to ground test measurements.

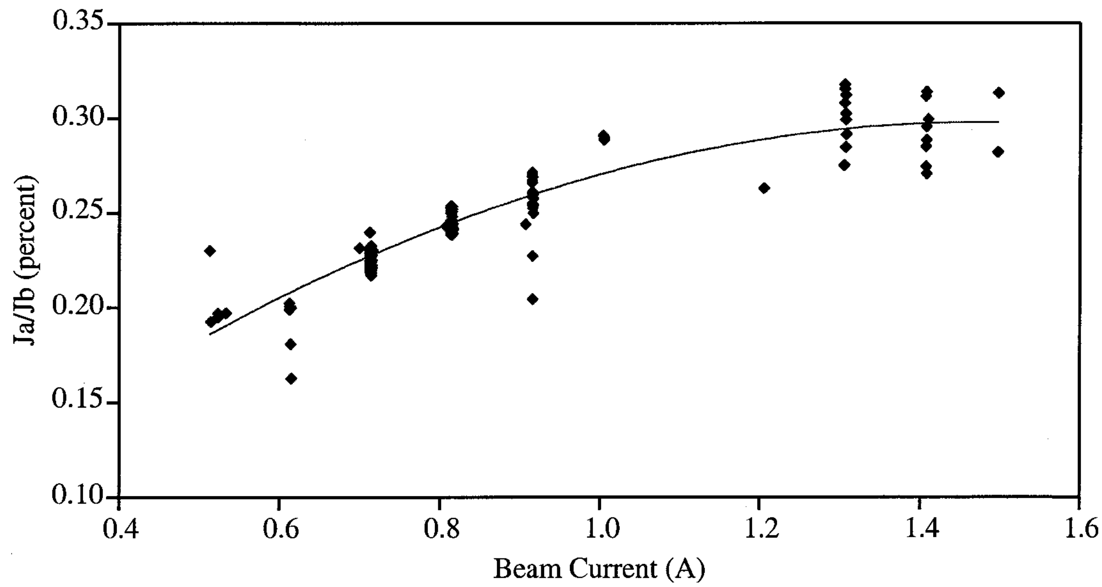

Figure 18: In-space ratio of accelerator grid impingement current to beam current.

were used to examine the effect of setpoint errors on the flight discharge parameters. The effects compete, and result in negligible changes in these parameters due to the small flow and beam current errors.

Ion Optics Performance

The ion optics appear to be performing very well so far in flight. The accelerator grid impingement current as a function of beam current is compared to ground test data in Fig. (17). The standard error of these measurements is about 0.03 mA. The data obtained in the ground test facilities are higher because they include a contribution from charge exchange reactions with residual tank gas. The initial impingement current levels in space are about 0.4 mA lower at 0.51 A and 1.7 mA lower at 1.5 A compared to preflight measurements in the JPL endurance test facility, which operates at pressure levels of 2–5$\times10^{-4}$ Pa (1.5–4$\times10^{-6}$ Torr) over the full throttle range. Accelerator grid erosion measurements obtained in long duration tests in this facility are therefore conservative. Data obtained in VF 5 at NASA GRC, which has a residual gas pressure about 3 times lower than that at JPL, show impingement currents which are about 0.4 mA greater than the initial space values. The impingement current increased slightly after the first 430 hours of operation and is now comparable to the lowest values measured on the ground. The ratio of impingement current to beam current is shown as a function of beam current in Fig. (18). This parameter, which is used in some probabilistic models of accelerator grid erosion [**?**, **?**, **?**] ranges from 0.19 percent at 0.51 A to 0.3 percent at 1.5 A with a standard deviation of 0.012 percent.

Approximately 171 high voltage faults have occurred during 7600 hours of engine operation (excluding those that occurred as a result of the initial grid short). There has been no evidence of electron backstreaming. The discharge loss has consistently increased slightly when the accelerator grid voltage is raised from -250 V after ignition to the throttle setpoint, which is the nominal behavior. This transition is monitored for decreases in the discharge loss, which could signal the loss of electron backstreaming margin.

Neutralizer Performance

The neutralizer power consumption has been 4–7 W lower than the BOL throttle table values due to a lower neutralizer keeper voltage, shown in Fig. (19). This power savings roughly compensates for a higher beam power demand due to the beam current offset. The voltage dropped by about 0.5 V over several days before many of these data were taken in IAT1. The IAT1 data show that at that point in the mission the keeper voltage was up to 2 V less than the ground test values. This difference is not yet understood. The voltage decreased with time over the first 4155 hours as it typically does in ground tests [**?**, **?**, **?**]. During operation at lower power levels over the last 3400 hours, however, the neutralizer keeper voltage appears to be about 1 V higher than it was during IAT2 at 1800 hours.

There is no instrumentation on the DS1 spacecraft that allows the true neutralizer coupling voltage to be easily determined. The voltage of neutralizer common with respect to the spacecraft ground is metered, and the behavior is shown in Fig. (20). The magnitude of this potential has increased slowly over the mission and is now about zero. To properly compare this with the ground measurements of coupling voltage, the spacecraft potential with respect to the ambient plasma must be known. Data from the Plasma Experiment for Planetary Exploration (PEPE), another experiment on DS1, suggest that the spacecraft potential is within 8 V of the ambient potential. The coupling voltage in space therefore appears to be less than 8 V. This is somewhat lower than that observed in ground tests, which is typically 12-14 V.

Mission Operations

Although the total thruster operating time so far has been orders of magnitude longer than that required by impulsive propulsion systems, the mission operations demands have been minimal. This is largely due to the successful implementation of a high degree of spacecraft autonomy.

XXX–integrate this stuff! These DCIU functions can be called with ground commands. In addition, the spacecraft can generate commands to the IPS to

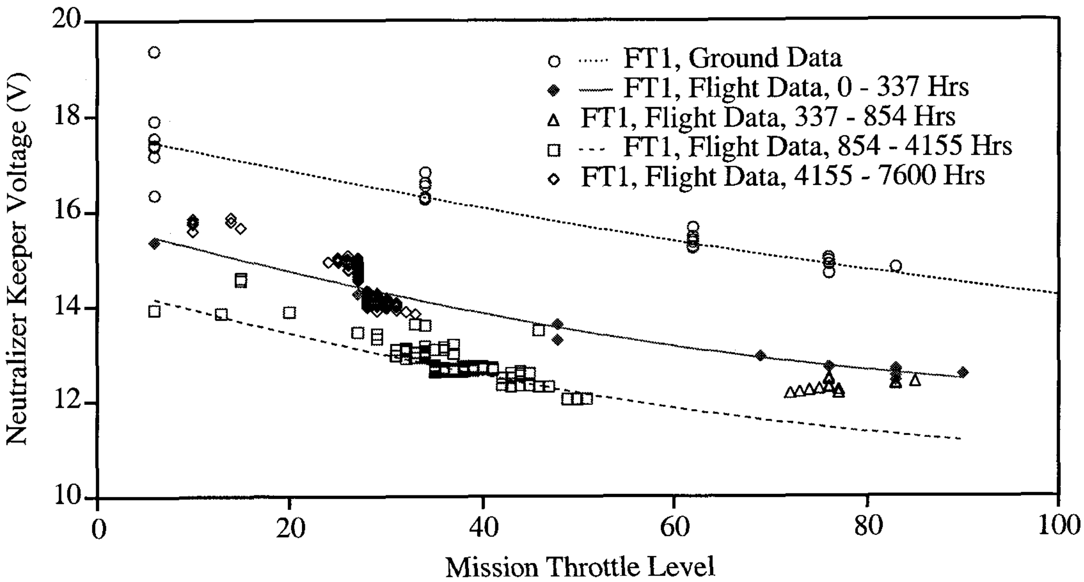

Figure 19: Neutralizer keeper voltage measured in space and in ground tests.

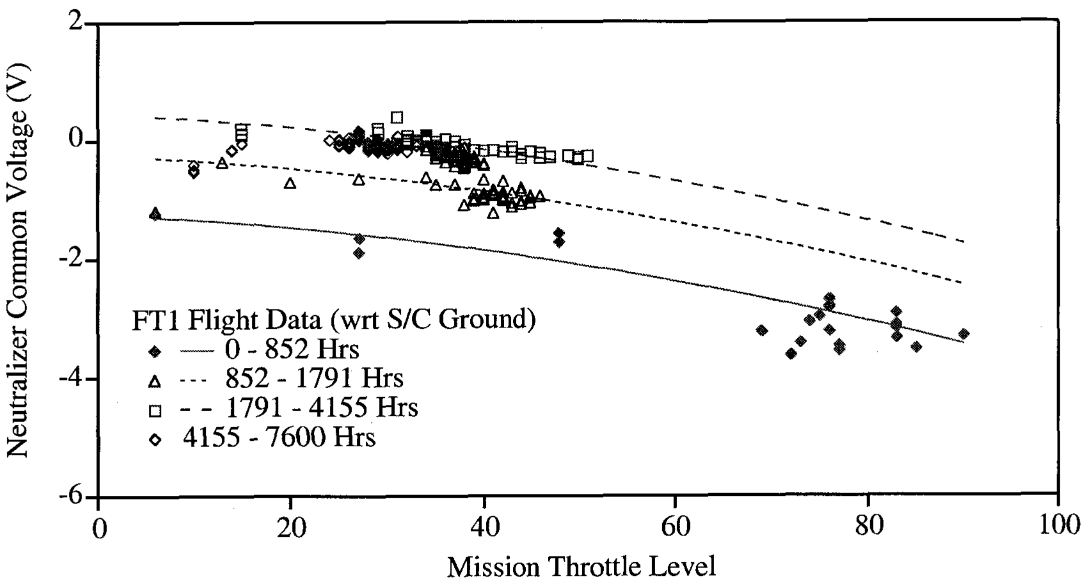

Figure 20: Neutralizer common voltage measured with respect to spacecraft ground in space.

perform certain operations. The IPS is throttled autonomously by the spacecraft to track the solar array output. DS1 also includes an autonomous system (AutoNav) to navigate the spacecraft to the next encounter target. This system contains an optimized trajectory that was computed on the ground and a catolog of ephemerides for a number of stars, asteroids, the planets and the DS1 target bodies. Periodically (one to three times per week) during a burn, the system automatically turns the spacecraft to optically observe the positions of a number of these bodies against the stellar background and calculates the spacecraft position. The heliocentric orbit is then determined and the trajectory propogated to the next target. Required course changes are generated by the maneuver design element and accomplished by varying the IPS thrust direction and duration. This technology dramatically reduces the need for mission operations support, as described below.

Autonomous navigation has significantly reduced the demands on the navigation and trajectory design teams and spacecraft control of the IPS relieves the ground controllers considerably. In the initial phase of the mission a number of propulsion engineers were involved in mission operations and validation. However, the final NBURNs have become sufficiently routine at this point that very little workforce is assigned to this area. The flight data dissemination and analysis has also been largely automated. During the initial, nearly continuous Deep Space Network coverage the spacecraft telemetry was displayed in real-time on a website that could be accessed by the flight team. Data are also stored in the JPL ground data system and automatic queries generate files that are sent via ftp to all flight team members. A series of macros in Igor Pro software are used to automatically load, analyze and plot these data. The success in reducing mission operations requirements with automation is an extremely significant result, because the fear of excessive operations costs has been a major barrier to the acceptance of ion propulsion for planetary missions. This flight demonstrates that mission operations costs for SEP-driven spacecraft are similar to those for conventional spacecraft or possibly less in cases where ion propulsion results in shorter trip times.

Conclusions

The test of ion propulsion on the Deep Space 1 mission has been extremely successful so far. All normal IPS functions and some of the fault recovery modes have been demonstrated over a total of 8000 hours of operation. The differences between system performance and engine operating characteristics in space and in ground tests have been very small. The thrust appears to be slightly lower than the calculated values at the higher power levels, and the PPU efficiency appears to be higher than the conservative values assumed in the throttling tables. Fully autonomous navigation and operation of the IPS have also been demonstrated, achieving the goal of minimizing the required ground support for low thrust propulsion systems. This flight validates ion propulsion technology for use on future interplanetary spacecraft and has provided a wealth of information for future mission and spacecraft designers.

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References

[1] M.D. Rayman and D.H. Lehman. Deep Space 1: NASA's First Deep-Space Technology Validation Mission. In 48^{th} *International Astronautical Congress*, Turin, Italy, 1997. IAF-97-Q.5.05.

[2] G.B. Ganapathi and C.S. Engelbrecht. Post-Launch Performance Characterization of the Xenon Feed System on Deep Space One. In 35^{th} *Joint Propulsion Conference*, Los Angeles, CA, 1999. AIAA-99-2273.

[3] J.J. Wang et al. Deep Space One Investigations of Ion Propulsion Plasma Interactions: Overview and Initial Results. In 35^{th} *Joint Propulsion Conference*, Los Angeles, CA, 1999. AIAA-99-2971.

[4] J. S. Sovey et al. Development of an Ion Thruster and Power Processor for New Millenium's Deep Space 1 Mission. In 33^{rd} *Joint Propulsion Conference*, Seattle, WA, 1997. AIAA-97-2778.

[5] J.A. Christensen, G. Benson, T.A. Bond, and M. Matranga. The NSTAR Ion Propulsion Subsystem for DS1. In 35^{th} *Joint Propulsion Conference*, Los Angeles, CA, 1999. AIAA-99-2972.

[6] J.E. Polk, J.R. Anderson, J.R. Brophy, V.K. Rawlin, M.J. Patterson, and J.S. Sovey. The Results of an 8200 Hour Wear Test of the NSTAR Ion Thruster. In 35^{th} *Joint Propulsion Conference*, Los Angeles, CA, 1999. AIAA-99-2446.

[7] J.E. Polk et al. A 1000-Hour Wear Test of the NASA NSTAR Ion Thruster. In 32^{nd} *Joint Propulsion Conference*, Lake Buena Vista, FL, 1996. AIAA-96-2717.

[8] J.E. Polk, J.R. Anderson, J.R. Brophy, V.K. Rawlin, M.J. Patterson, and J.S. Sovey. The Effect of Engine Wear on Performance in the NSTAR 8000 Hour Ion Engine Endurance Test. In 33^{rd} *Joint Propulsion Conference*, Seattle, WA, 1997. AIAA-97-3387.

[9] M. Patterson, T. Haag, V. Rawlin, and M Kussmaul. NASA 30 cm Ion Thruster Development Status. In 30^{th} *Joint Propulsion Conference*, Indianapolis, 1994. AIAA-94-2849.

[10] T.P. McElrath, D. Han, and M.S. Ryne. Radio Navigation Of Deep Space 1 During Ion Propulsion Usage. In 15^{th} *International Symposium of Spaceflight Dynamics*, Biarritz, France, 2000. MS00/57.

[11] V. Rawlin, M. Patterson, and R. Becker. Thermal Environmental Testing of NSTAR Engineering Model Ion Thrusters. In 25^{th} *International Electric Propulsion Conference*, Cleveland, OH, 1997. IEPC 97-051.

[12] J.R. Anderson, K.D. Goodfellow, J.E. Polk, and R.F. Shotwell. The Results of an 8200 Hour Wear Test of the NSTAR Ion Thruster. In 35^{th} *Joint Propulsion Conference*, Los Angeles, CA, 1999. AIAA-99-2857.

[13] J.E. Polk, N.R. Moore, L.E. Newline, J.R. Brophy, and D.H. Ebbeler. Probabilistic Analysis of Ion Engine Accelerator Grid Life. In 23^{rd} *International Electric Propulsion Conference*, Seattle, WA, 1993. AIAA-93-176.

[14] J.R. Anderson, J.E. Polk, and J.R. Brophy. Service Life Assessment for Ion Engines. In 25^{th} *International Electric Propulsion Conference*, Cleveland, OH, 1997. IEPC 97-049.

[15] J.R. Brophy, C.E. Garner, J.E. Polk, and J. Weiss. The Ion Propulsion System on NASA's Space Technology 4/Champollion Comet Rendezvous Mission. In 35^{th} *Joint Propulsion Conference*, Los Angeles, CA, 1999. AIAA-99-2856.